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NAVAIRDEVCEN GRAPHITE-EPOXY COMPOSITE WING FOR BQM-34E: STATIC TEST RESULTS

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A graphite-epoxy composite wing for the BQM-34E unmanned supersonic aerial target vehicle has been successfully static tested to 100 percent design limit load at the NAVAIRDEVCEN.

S U M M A R Y

A graphite-epoxy composite wing for the BQM-34E unmanned supersonic aerial target vehicle has been successfully static tested to 100 percent design limit load at the NAVAIRDEVCEN (Naval Air Development Center). Test procedures and results are described, and comparisons between analytical predictions and recorded values are presented. Good correlation between analysis and test was observed. On the basis of this test, the composite wing is considered statically qualified for flight.

I N T R O D U C T I O N

A graphite-epoxy wing for the BQM-34E unmanned supersonic target vehicle was designed and fabricated by the NAVAIRDEVCEN. The completed component was required to undergo vibration testing and a static proof test to 100 percent design limit load to ensure structural adequacy prior to flight testing.

The existing metal wing is a symmetric modified NACA 65-003 airfoil with leading edge sweep of 53 degrees, moderate aspect ratio, and low thickness ratio. The metal wing is constructed of chem-milled stainless-steel skins bonded to a full depth aluminum honeycomb core. Each semi-span is bolted to an aluminum center section and there are separate leading and trailing edge pieces.

The design of the composite wing placed emphasis upon reducing the weight and simplifying the manufacturing process while maintaining the original airfoil shape and planform. Since the original design was flutter critical, the composite wing had to satisfy both the strength and flutter criteria of the original wing. A more detailed discussion of the design and analysis of the composite wing can be found in references (a) and (b).

The basic construction of the composite wing consists of optimized laminated graphite-epoxy skins bonded to an aluminum honeycomb core. The aluminum honeycomb core consists of 4.5 lbs./ft.^{3} and 6.1 lbs./ft.^{3} density core with local reinforcement of 8.1 lbs./ft.^{3} and 23 lbs./ft.^{3} density core. The graphite-epoxy skins vary in thickness from 30 plies at the center section to five plies near the leading and trailing edges. The leading edge is made from chopped graphite fibers which are packed into a tool and cured to give the appropriate leading edge shape. There is no separate trailing edge piece. Fiberglass conduit runs through the core to carry wires to the tip antenna. The laminate adjacent to the aft wing-fuselage attachment bolts is highly stressed. In addition, space limitations require placement of the holes very close to the free edge of the skins. Accordingly, the graphite-epoxy skin is locally reinforced with titanium. The titanium plate is spliced to the laminate through specially designed step joints. Graphite channels were fabricated and bonded in place in the area of the root chord and as closeouts for the center section. The composite wing assembly, including the center section, is an integral piece, thus

significantly reducing the total number of parts and simplifying fabrication. A more detailed discussion of the composite wing manufacture can be found in reference (b). A reduction of 54 percent of the wing weight was achieved while the strength and stiffness requirements were still met.

As part of the structural qualification of the composite wing, a vibration survey was performed prior to static testing, and the results are reported in reference (c). Following the 100 percent design limit load static proof test, reported herein, the composite wing was transported to the NAVMISCEN (Naval Missile Center, Point Mugu, California) for flight testing on the Pacific Missile Range. The results of the flight test are reported in reference (d).

DESCRIPTION OF TEST SPECIMEN

The unpainted wing as received weighed 70 pounds. The final finished and painted wing weighs 72 pounds. The graphite skins had a somewhat rough finish from the protective nylon peel ply which was removed before testing. Since the skin of the wing was locally damaged during the final bonding operation, two circular repair patches were bonded over opposite surfaces in the damaged area. The repair patches each consist of two layers of fiberglass and two layers of thin titanium sheet. Figure 1 is a photograph of the completed wing with the repair patch on the left semispan below the NAVAIRDEVCEN logo. Figure 2 is a dimensioned sketch of the wing planform, showing the reference coordinate system which will be used in the remainder of the report.

TEST CONDITION

The wing was tested to the critical loading condition designated 2SD02 in reference (e). This condition results from a subsonic ($M = 0.614$) symmetric 5g maneuver at sea level. Shear, moment and torque curves used in the static test as compared to the theoretical curves are shown in Figures 3, 4, and 5 respectively.

STATIC TEST SET-UP

The BQM-34E graphite-epoxy wing was mounted on a test frame of 12-inch wide-flange erecto beams using an adapter plate. The wing was bolted to the adaptor plate using standard aircraft bolts through the 10 wing-to-fuselage attachment bolt holes. Steel spacers .400 inch thick and 5/8 inch in diameter were used between the wing and the plate to simulate aluminum bosses which exist in the actual aircraft installation. This assembly was then bolted to the beam in an inverted position to facilitate testing. Loading was accomplished through the use of 58 tension pads distributed over the compression surface of the wing, with an additional 12 compression pads along the leading edge on the tension surface. Figure 6 shows the tension and compression pad lay-out while

Table I gives the coordinates and the limit load of each pad. Loads were distributed to each pad from a single pull point through the use of a whiffletree lever system, with the compression pads being loaded from the same whiffletree via C-shaped brackets. Test loads were applied through the use of a hand-operated hydraulic actuator and monitored by a strain-gage load cell. Average stresses for all tension pads were kept below 15 psi. The overall test set-up is shown in Figure 7.

The composite wing was instrumented with 58 resistance strain gages bonded to the outer surface and 14 potentiometric deflection transducers attached by wires to wooden blocks bonded to the lower surface. Figure 8 shows the location and orientation of the strain gages and Table II gives the actual coordinates of each gage which are located in critically stressed areas. Figure 9 and Table III give the same information for the deflection transducers. Deflection and strain data were recorded using a high speed digital data acquisition system capable of recording its values on paper tape at the rate of 20 channels per second.

STATIC TEST PROCEDURES

In the initial test run, loads were applied in increments of 10 percent of limit load to 50 percent limit load, then in five percent increments to 75 percent limit load, and then released. In the subsequent run, loads were applied in 20 percent increments to 80 percent limit load, then in five percent increments to 100 percent limit load, and then released. After each load above 75 percent limit load a visual inspection of the wing and the test setup was made. Strains and deflections were recorded at each load increment. Calibration and zero readings were taken at five percent limit load before and after each run.

TEST RESULTS

The wing sustained the design limit loads without failure. Visual inspections during and after the test revealed no apparent structural damage. No noises were heard during the test, and loading was smooth and uniform. Plots of strain and deflection versus load showed no discontinuities which might be indicative of incipient structural failure.

Typical plots of strain versus percent of limit load for the final run are shown in Figures 10 and 11. Gage identifcation numbers correspond to those shown in Figure 8. All data plotted has been corrected to zero load via extrapolation. It should be noted that the load versus strain plots are linear throughout the test range. This is typical of all data recorded.

Recorded strains are listed in Table III. Correlation of the predicted strains from the NASTRAN structural computer analysis (reference

(a)), with those recorded during the test is considered to be good. Figures 12 and 13 present comparisons of analytically predicted stresses versus experimentally measured stress along chordwise sections A-A and B-B, shown in Figure 8.

Experimental deflections recorded are listed in Table III. Typical plots of deflection versus percent of limit load are presented in Figures 14 and 15. Transducer numbers correspond to those given in Figure 9. Figures 16, 17 and 18 are plots of the analytically predicted deflections and their corresponding experimental measurements along the leading edge, the 56-percent chordline, and the trailing edge, respectively.

CONCLUSIONS

The static test verified the structural adequacy of this graphite-epoxy composite BQM-34E wing. The strains and deflections measured are generally slightly lower than those calculated by the finite element structural analysis. Since they are in the conservative direction, it is concluded that the composite wing design is structurally adequate for flight.

REFERENCES

(a) Neu, T. F. and Huang, S. L.: NAVAIRDEVCEN Graphite-Epoxy Composite Wing for BQM-34E: Stress and Vibration Analysis. NAVAIRDEVCEN Report No. 73235-30, 15 Nov 1973.

(b) NAVAIRDEVCEN Graphite-Epoxy Composite Wing for BQM-34E: Manufacturing Procedure. To be published.

(c) Somoroff, A. R., and Rubin, H.: NAVAIRDEVCEN Graphite-Epoxy Composite Wing for BQM-34E: Aeroelastic Analysis. NAVAIRDEVCEN Report No. 73233-30, 12 Nov 1973.

(d) NAVAIRDEVCEN Graphite-Epoxy Composite Wing for BQM-34E: Flight Test Results. To be published.

(e) Krzyzanowsici, A. and Lambert, C. G.: Wing Structural Analysis Report for BQM-34 Supersonic Aerial Target. Teledyne Ryan Aeronautical Report Number TRA 16642-12, 6 Jan 1971.

TABLE I

TENSION PAD LOCATION AND LOADS (SEMISPAN)

PAD NO.	PAD LOCATIONS* X	Y	100% DLL LBS.	PAD NO	PAD LOCATIONS* X	Y	100% DLL LBS.
1	-22.81	42.1	357	20	-22.36	22.25	45
2	-29.41	42.1	60	22	16.91	14.9	450
3	-33.81	42.1	34	23	11.29	14.9	203
4	-12.21	37.0	264	24	5.79	14.9	110
5	-16.31	37.0	57	25	.29	14.9	100
6	-21.81	37.0	48	26	-4.71	14.9	80
7	-26.61	37.0	24	27	-9.71	14.9	60
8	-30.31	37.0	10	28	-14.90	14.9	40
9	-2.66	29.75	447	29	-20.10	14.9	20
10	-9.06	29.75	242	30	25.70	7.5	421
11	-14.56	29.75	60	31	18.69	7.5	130
12	-19.56	29.75	40	32	13.19	7.5	125
13	-25.71	29.75	23	33	7.69	7.5	120
15	6.16	22.25	472	34	2.69	7.5	110
16	-1.56	22.25	250	35	-2.31	7.5	100
17	-7.06	22.25	100	36	-7.50	7.5	60
18	-12.06	22.25	70	37	-16.60	7.5	50
19	-17.04	22.25	50				

*Coordinates Refer to Axes Shown on Figure 2.

TABLE II

STRAIN GAGE LOCATION

GAGE NUMBER	WING SURFACE TENSION OR COMPRESSION	GAGE LOCATION* X	Y	GAGE** ORIENTATION (DEGREES)	EXPERIMENTAL μ-STRAIN AT 100% DLL
0	T	-11.7	25.0	0	1627
1	T	-11.7	25.0	90	- 849
2	T	-11.7	25.0	45	- 22
3	C	-11.7	25.0	0	-1784
4	C	-11.7	25.0	90	854
5	C	-11.7	25.0	45	- 65
6	T	- 1.8	20.0	0	-1675
7	T	- 1.8	20.0	90	- 796
8	T	- 1.8	20.0	45	- 30
9	T	-11.7	20.0	0	1741
10	T	-11.7	20.0	90	- 757
11	T	-11.7	20.0	45	58
12	C	-11.7	20.0	0	-1835
13	C	-11.7	20.0	90	712
14	C	-11.7	20.0	45	- 83
15	T	-20.9	20.0	0	1011
16	T	-20.9	20.0	90	- 335
17	T	-20.9	20.0	45	97
18	T	8.8	15.0	0	1140
19	T	8.8	15.0	90	--

* Coordinates refer to axes shown on Figure 2.

**Gage orientation with respect to local material axis (counterclockwise) positive).

TABLE II (CON'T)

GAGE NUMBER	WING SURFACE TENSION OR COMPRESSION	GAGE LOCATION* X	Y	GAGE** ORIENTATION (DEGREES)	EXPERIMENTAL μ—STRAIN AT 100% DLL
20	T	8.8	15.0	45	9
21	T	- 2.7	15.0	0	1545
22	T	- 2.7	15.0	90	--
23	T	- 2.7	15.0	45	24
24	T	-14.1	15.0	0	1109
25	T	-14.1	15.0	90	- 226
26	T	-14.1	15.0	45	44
27	T	- 2.0	10.0	0	1524
28	T	- 2.0	10.0	90	- 706
29	T	- 2.0	10.0	45	--
30	C	- 2.0	10.0	0	-1621
31	C	- 2.0	10.0	90	681
32	C	- 2.0	10.0	45	101
33	T	20.7	4.8	0	517
34	T	20.7	4.8	90	- 56
35	T	20.7	4.8	45	618
36	T	10.3	4.8	0	985
37	T	10.3	4.8	90	- 415
38	T	10.3	4.8	45	580
39	C	10.3	4.8	0	-1082
40	C	10.3	4.8	90	630

* Coordinates refer to axes shown on Figure 2.
**Gage orientation with respect to local material axis (counterclockwise) positive).

TABLE II (CON'T)

GAGE NUMBER	WING SURFACE TENSION OR COMPRESSION	GAGE LOCATION* X	GAGE LOCATION* Y	GAGE** ORIENTATION (DEGREES)	EXPERIMENTAL μ—STRAIN AT 100% DLL
41	C	10.3	4.8	45	- 668
42	T	4.0	- 2.0	0	1517
43	T	4.0	- 2.0	90	- 308
44	T	4.0	- 2.0	45	1323
45	T	- 1.8	10.0	0	1529
46	T	- 1.8	10.0	90	964
47	T	- 1.8	10.0	45	- 144
48	C	- 1.8	10.0	0	-1699
49	C	- 1.8	10.0	90	- 717
50	C	- 1.8	10.0	45	- 65
51	T	-11.7	25.0	0	1594
52	T	-11.7	25.0	90	- 799
53	T	-11.7	25.0	45	- 7
54	C	-11.7	25.0	0	-1861
55	C	-11.7	25.0	90	957
56	C	-11.7	25.0	45	- 175
57	T	11.2	- 4.0	0	1261
58	C	11.2	- 4.0	0	-1390

* Coordinates refer to axes shown on Figure 2.
**Gage orientation with respect to local material axis (counterclockwise) positive).

TABLE III

DEFLECTION TRANSDUCER LOCATIONS

TRANSDUCER NUMBER	TRANSDUCER LOCATIONS* X	TRANSDUCER LOCATIONS* Y	EXPERIMENTAL DEFLECTIONS AT 100% DLL (INCHES)
1	-38.6	43.6	5.02
2	-30.9	43.6	4.19
3	-19.8	43.6	3.68
4	-22.7	35.0	3.11
5	-29.5	25.0	2.31
6	-14.3	25.0	1.55
7	5.1	25.0	.78
8	-22.5	10.0	.64
9	- 1.5	10.0	.38
10	25.0	10.0	.07
11	1.0	- 9.0	- .02
12	-38.6	-61.6	4.93
13	-30.9	-61.6	4.60
14	-19.8	-61.6	3.77

*Coordinates Refer to Axes Shown on Figure 2.

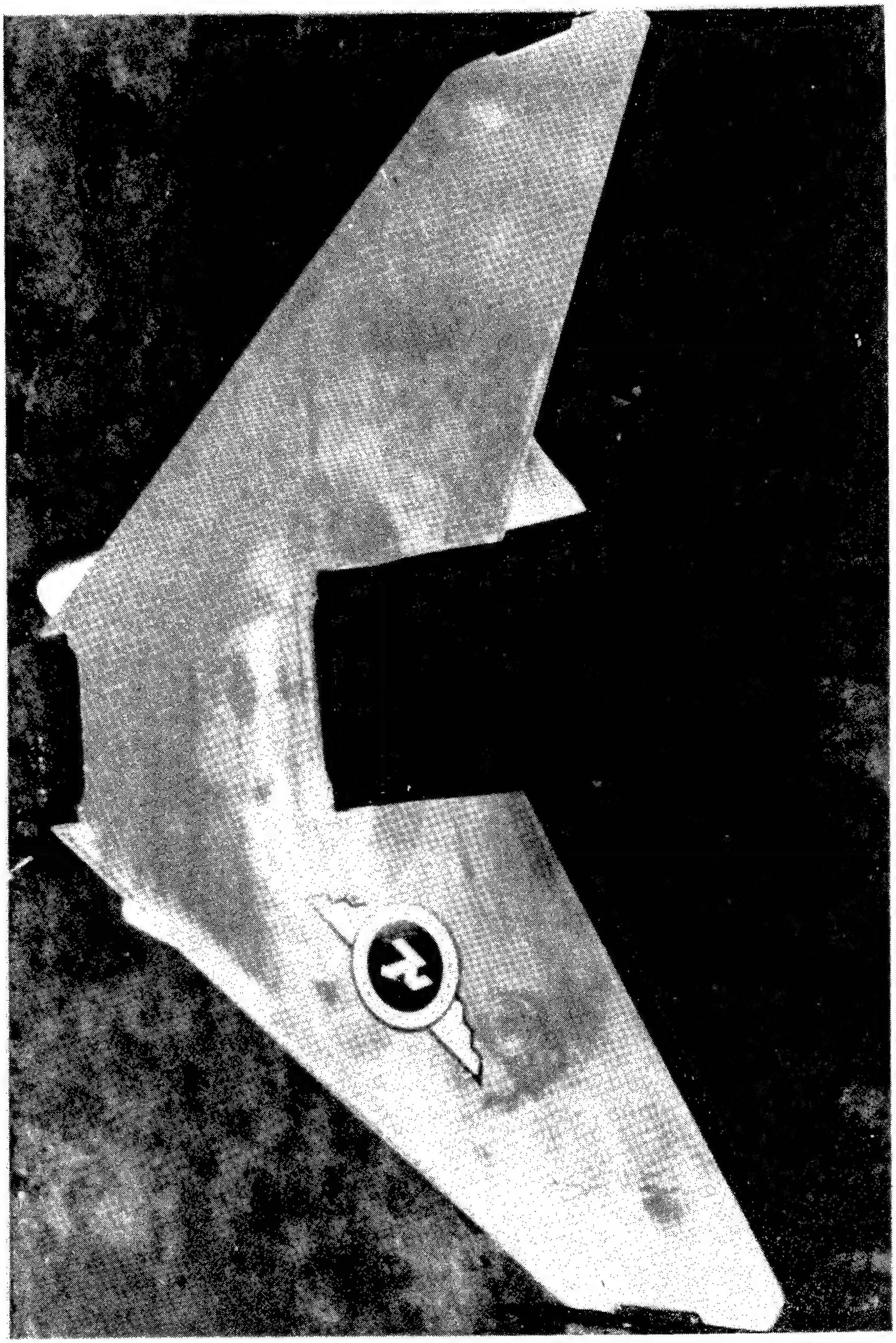


FIGURE 1 - BQM-34E GRAPHITE-EPOXY WING

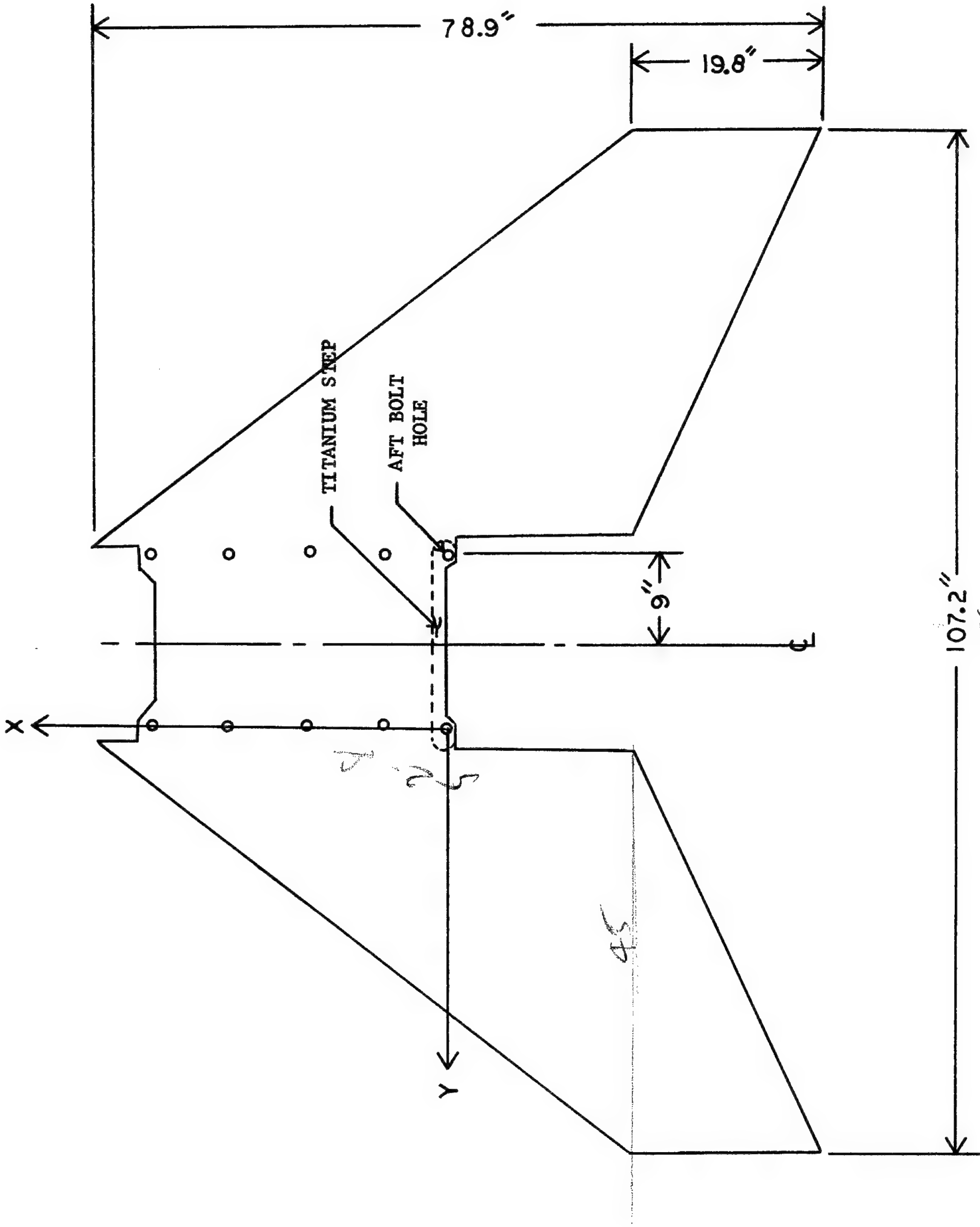


FIGURE 2 - BQM-34E WING PLANFORM

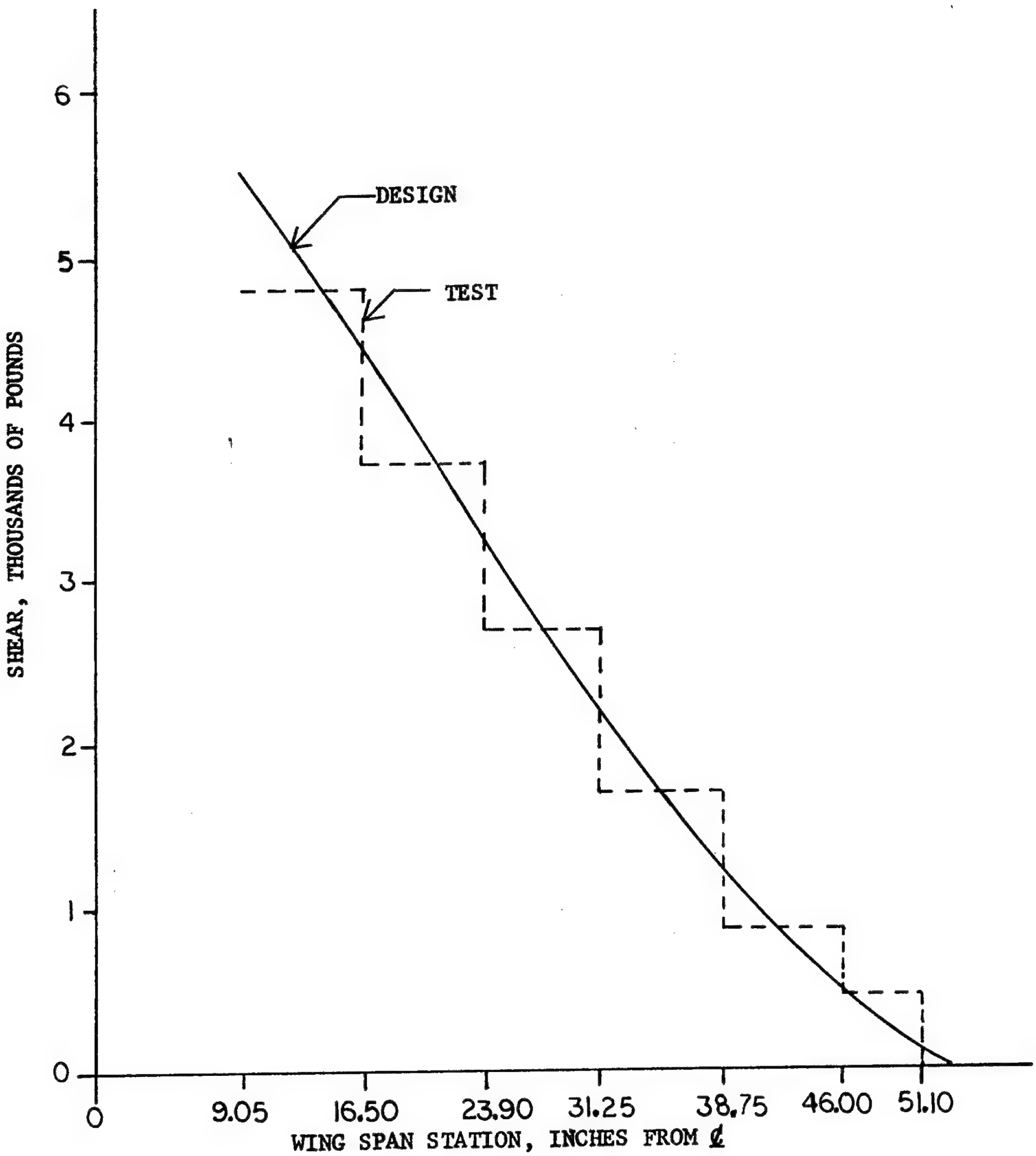


FIGURE 3 - WING LIMIT SHEAR

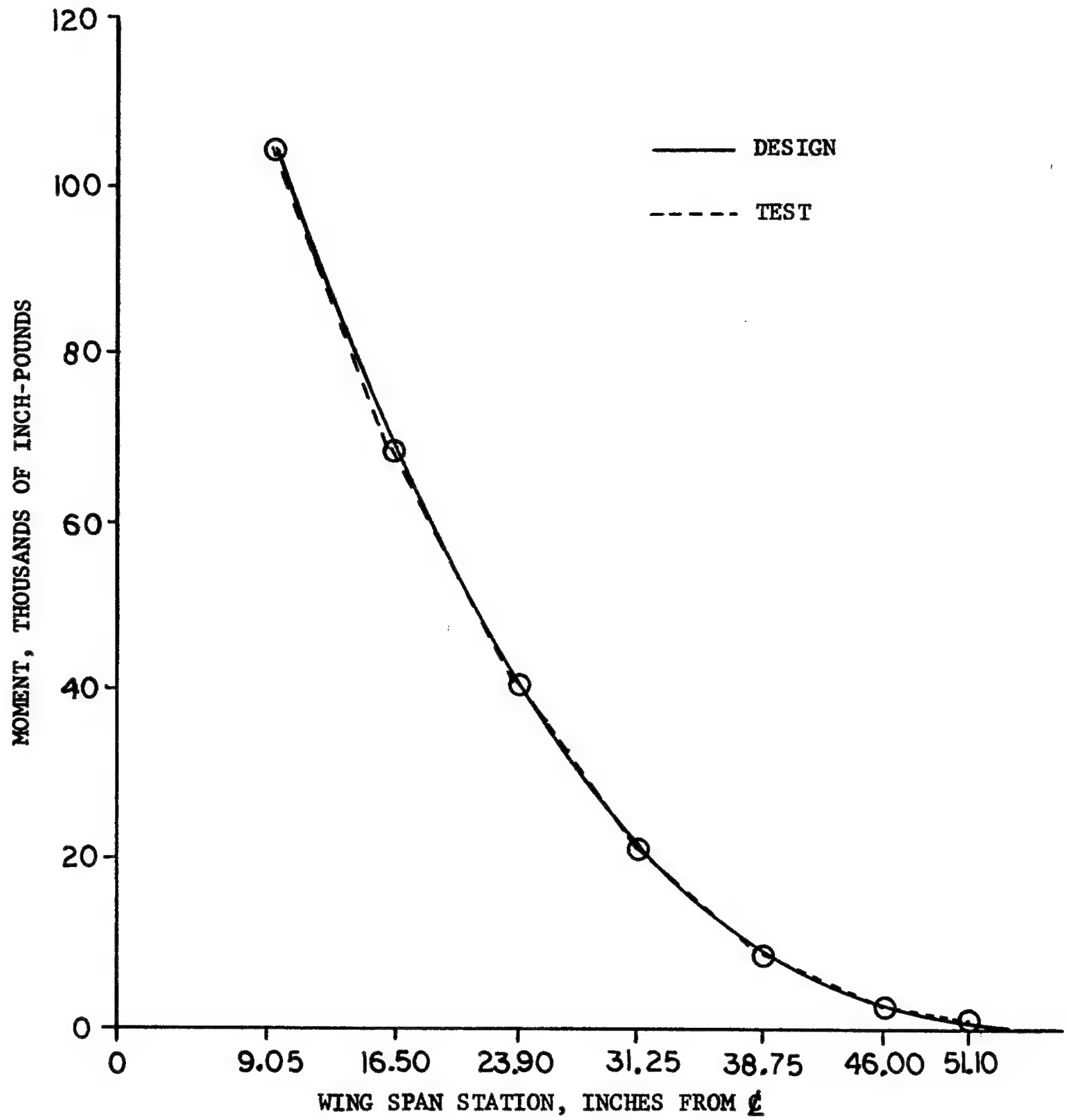


FIGURE 4 - WING LIMIT BENDING MOMENT

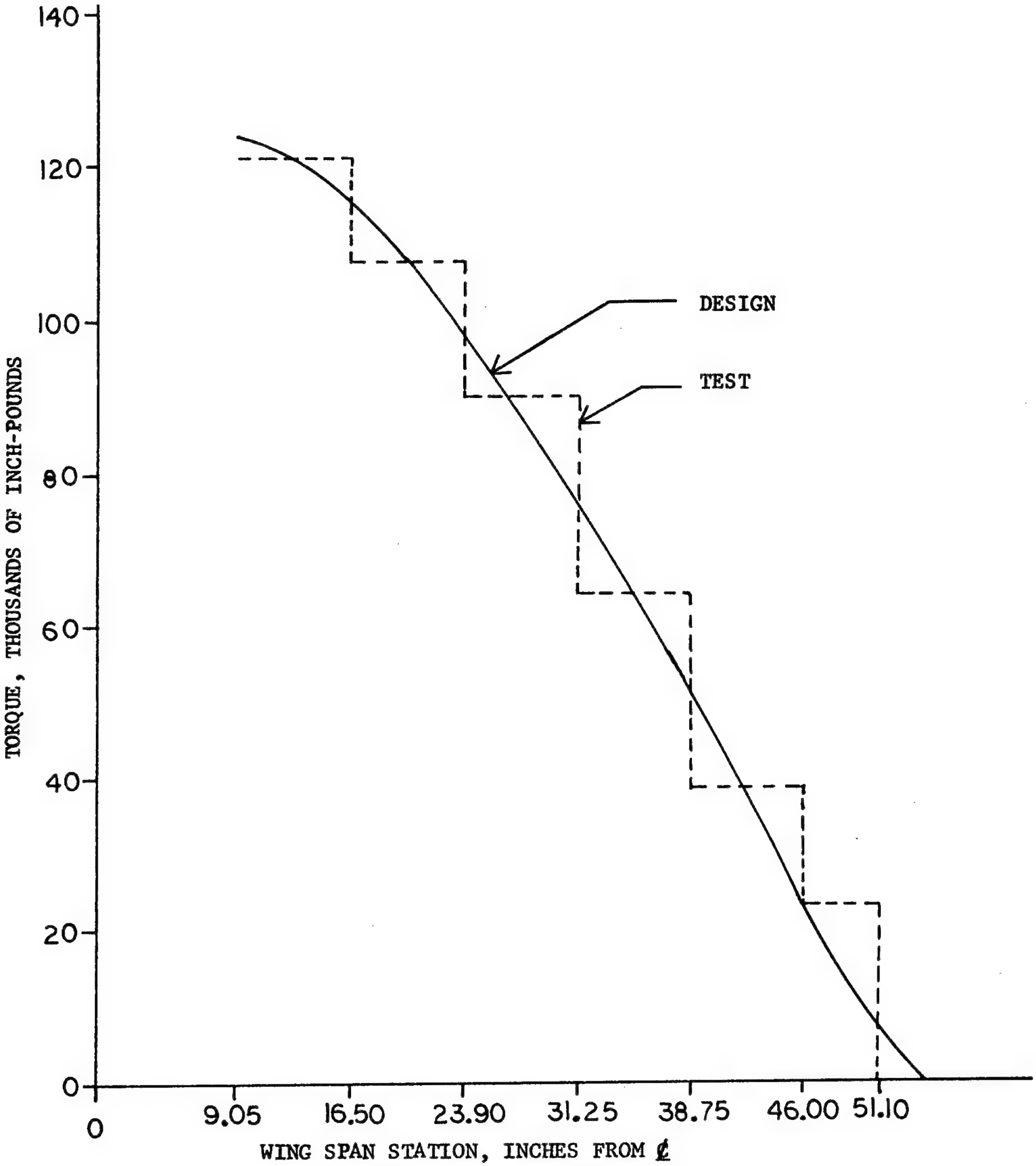


FIGURE 5 - WING TORQUE ABOUT $.40_{CR}$

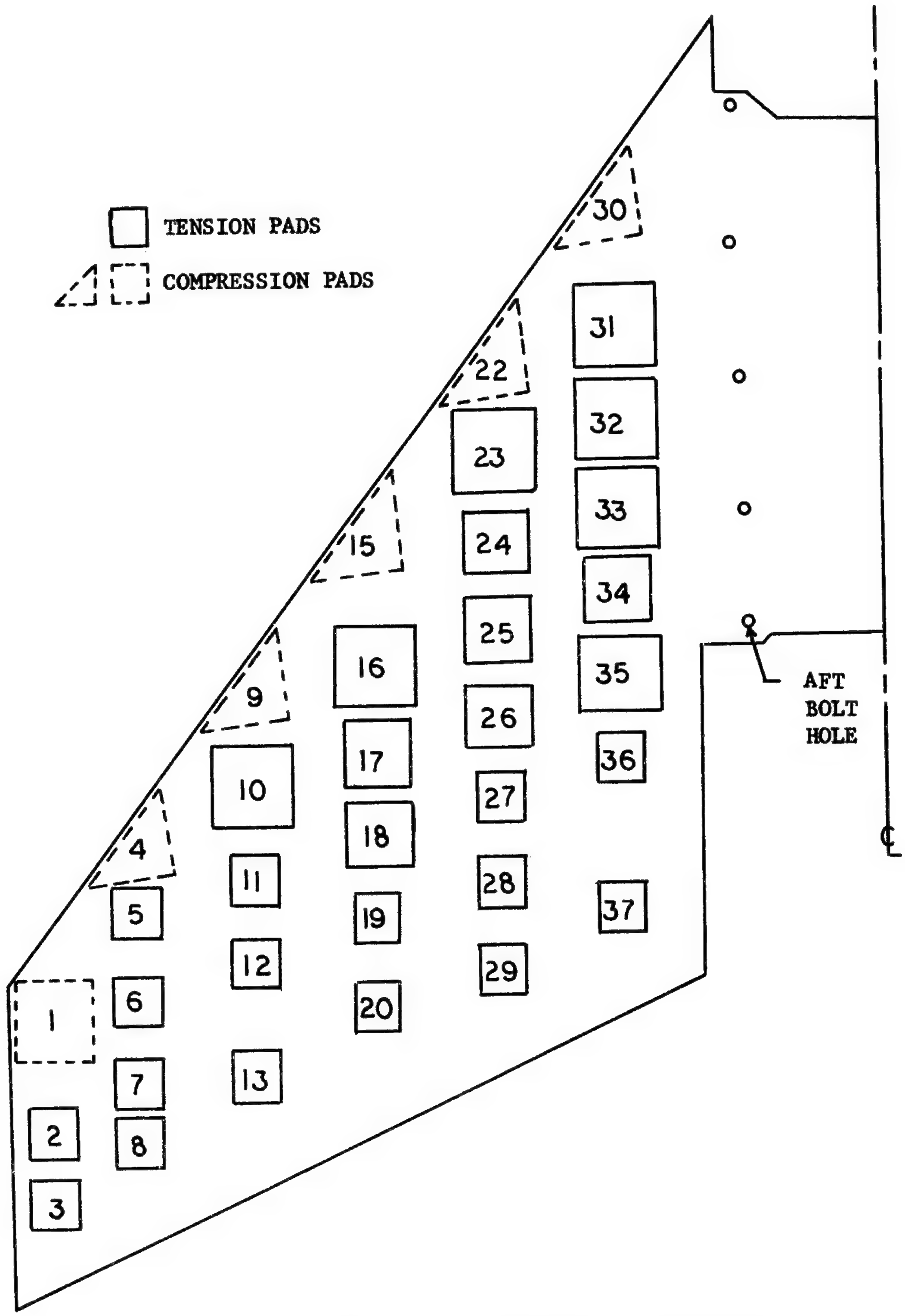


FIGURE 6 - TENSION PAD ARRANGEMENT (SEMISPAN)

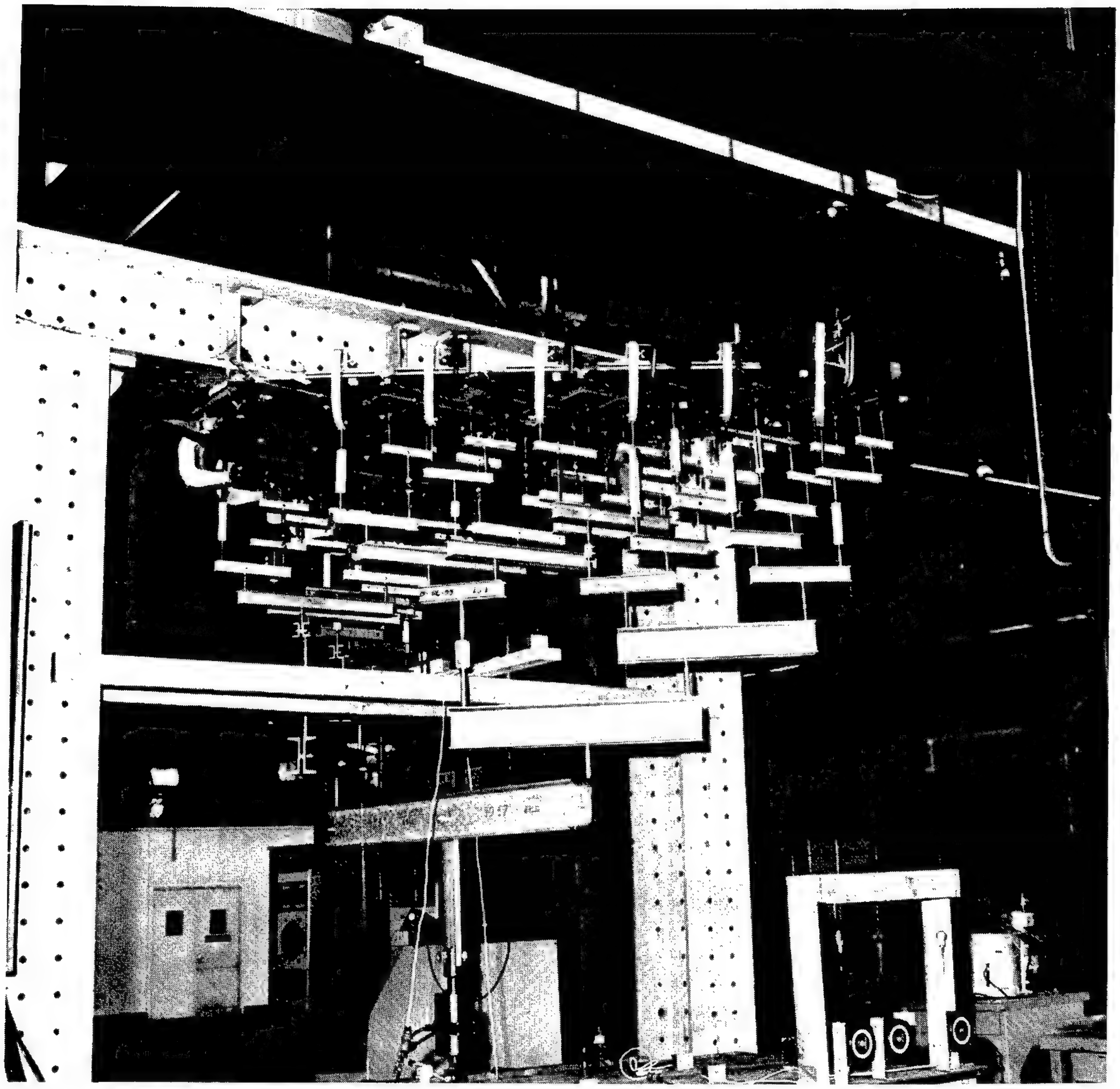
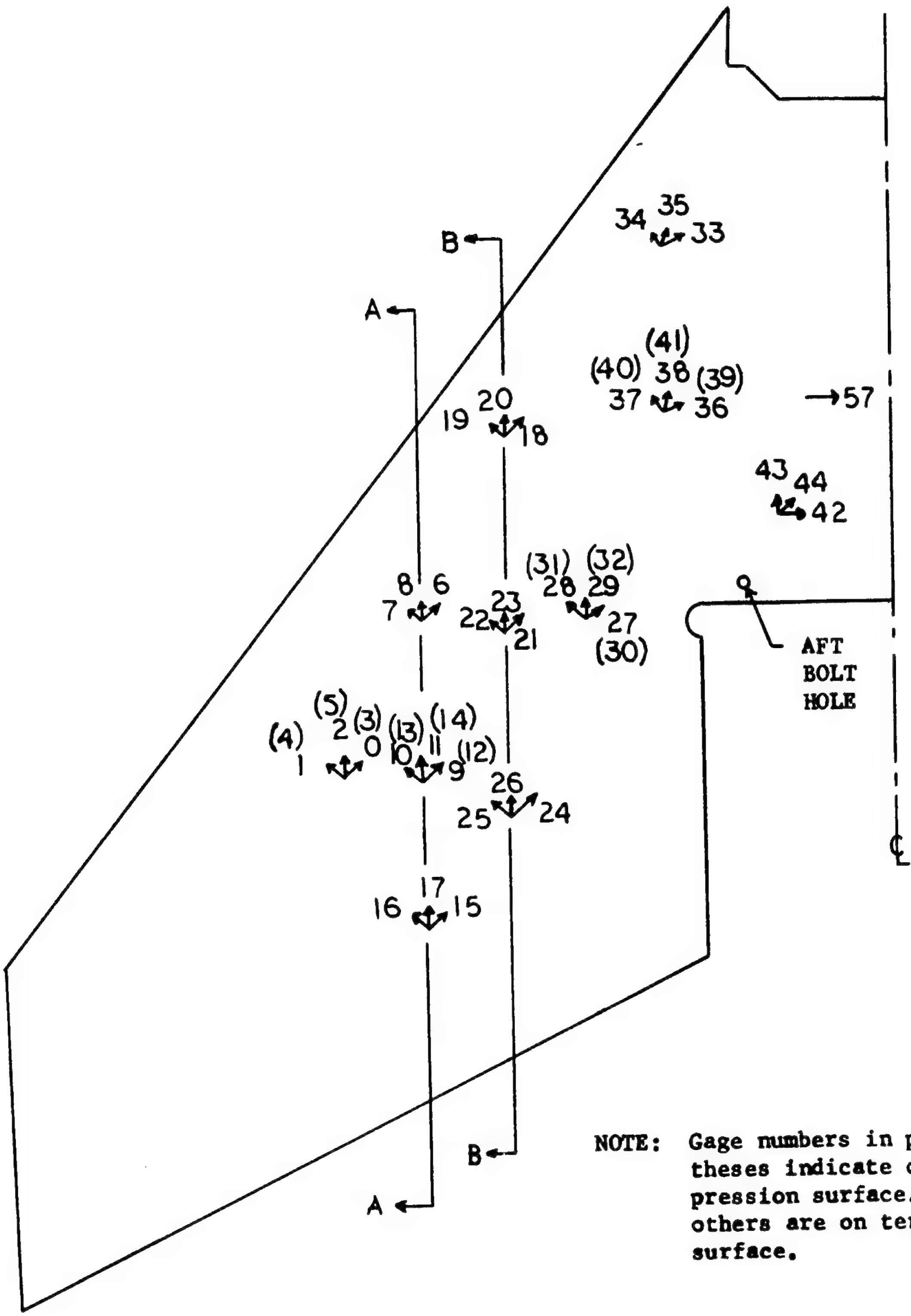


FIGURE 7 - OVERALL VIEW OF STATIC TEST SETUP



NOTE: Gage numbers in parentheses indicate compression surface. All others are on tension surface.

FIGURE 8 - STRAIN GAGE LAYOUT

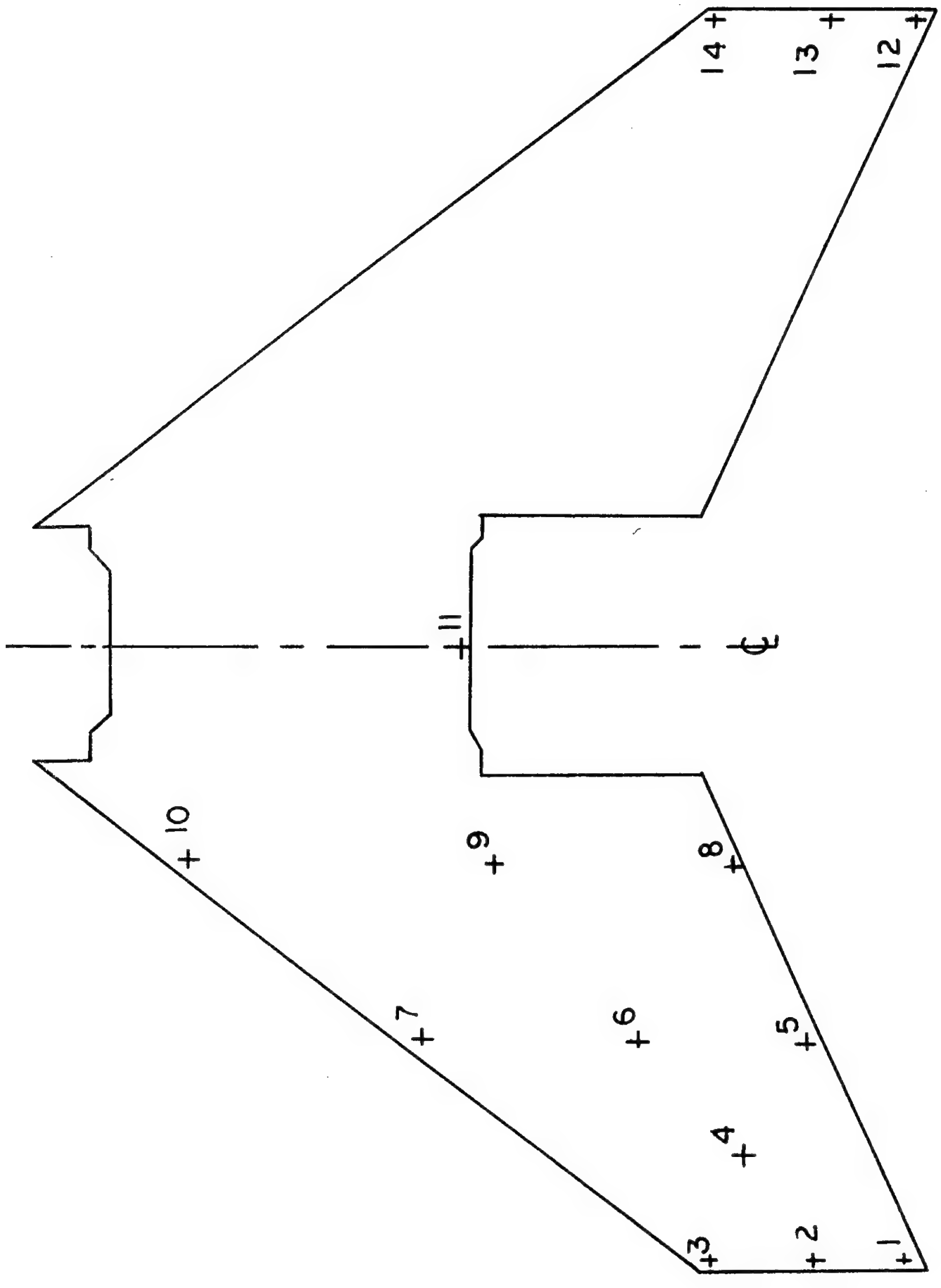


FIGURE 9 - DEFLECTION TRANSDUCER LAY-OUT

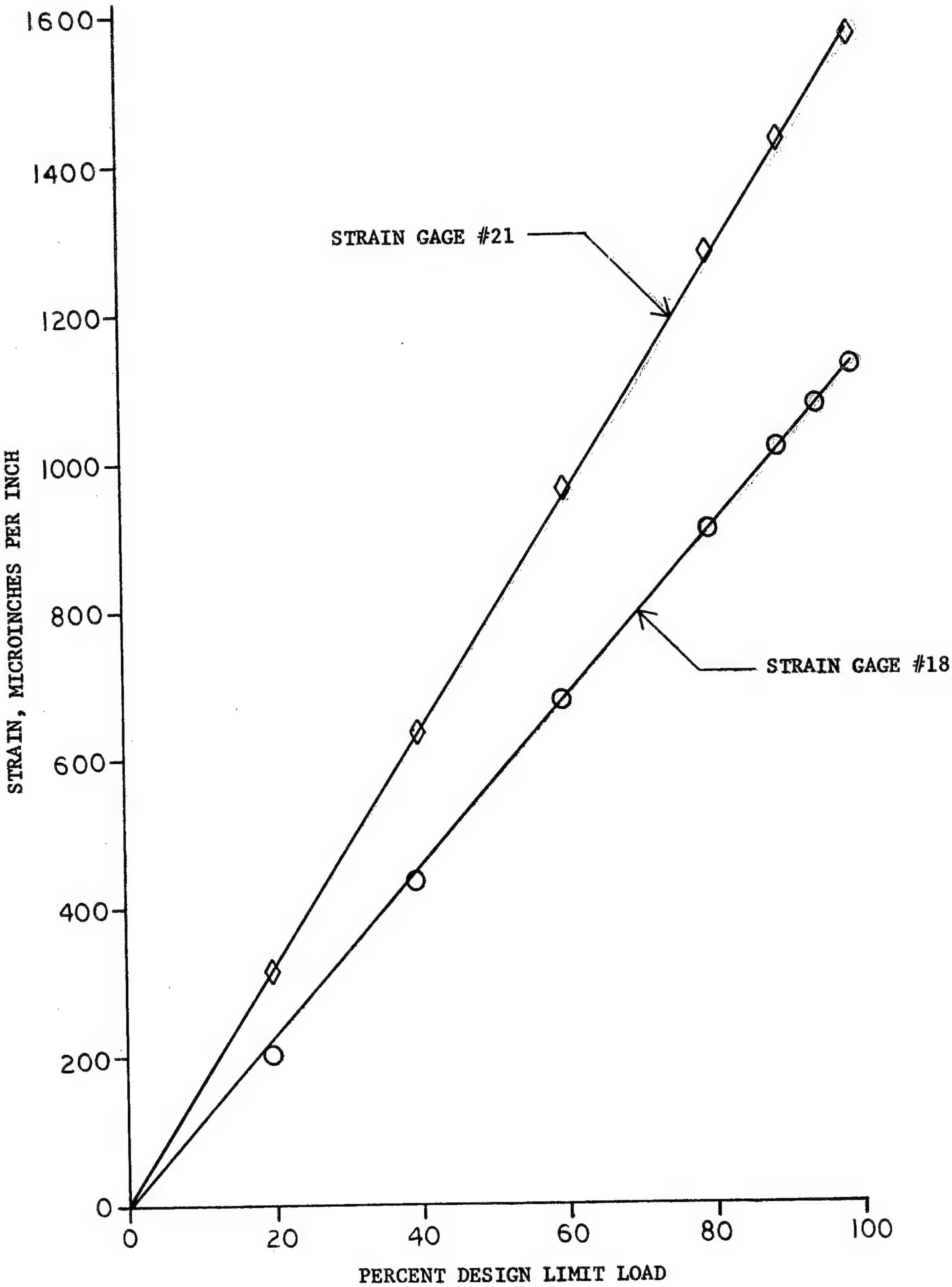


FIGURE 10 - TYPICAL RECORDED STRAIN VS PERCENT LIMIT LOAD, GAGES 18 AND 21

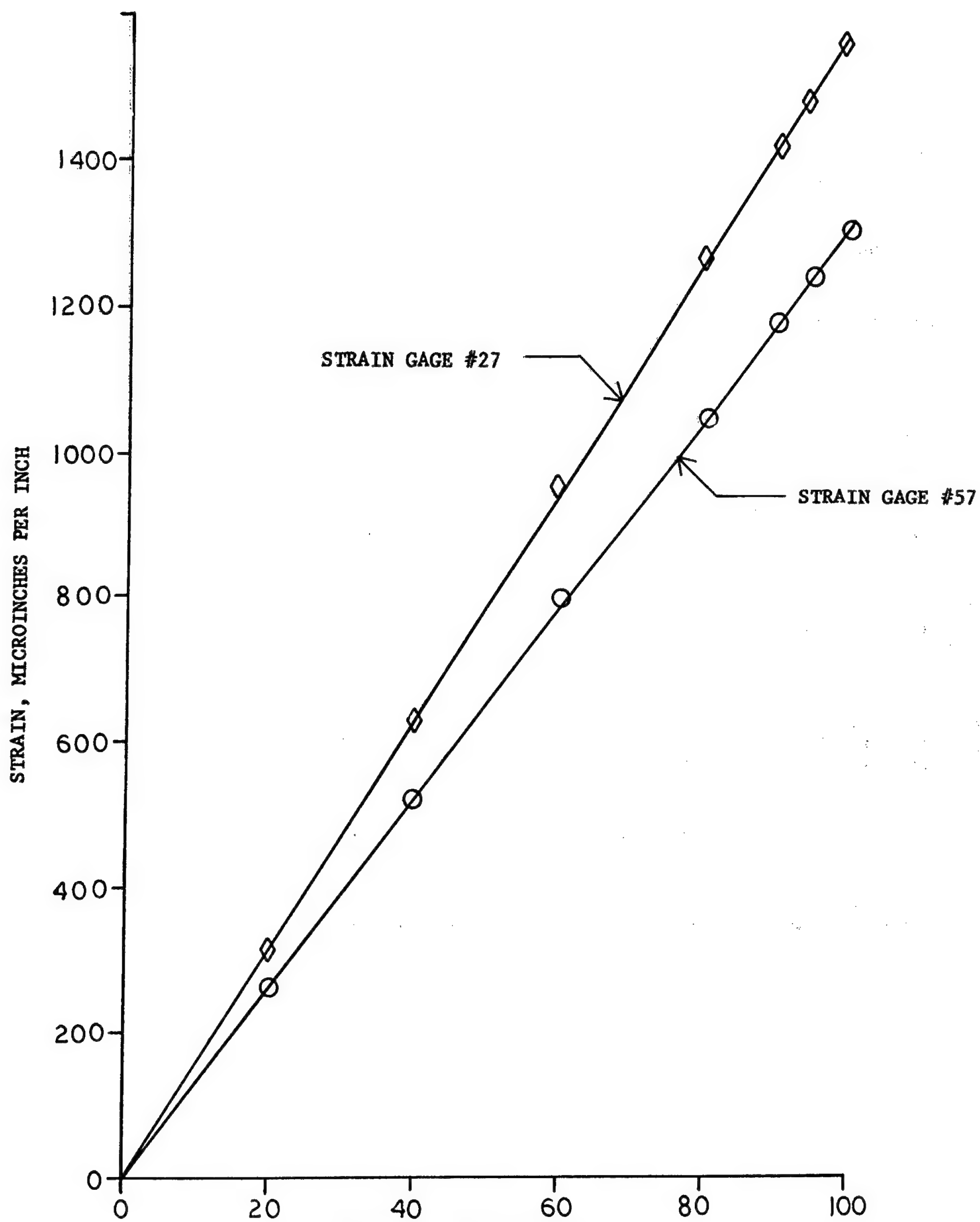


FIGURE 11 - TYPICAL RECORDED STRAIN VS PERCENT LIMIT LOAD, GAGES 27 AND 57

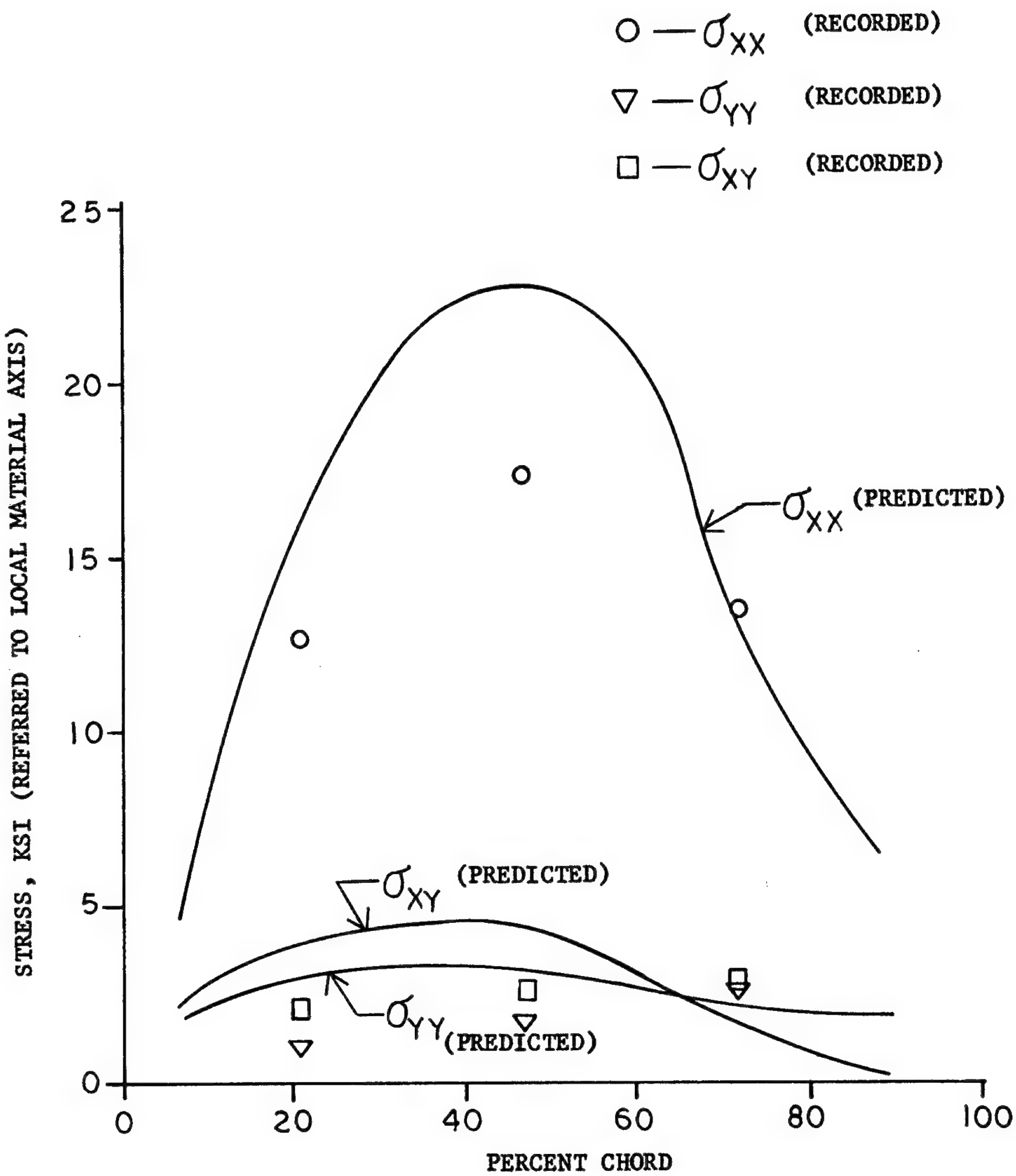


FIGURE 12 - COMPARISON OF PREDICTED AND RECORDED STRESSES IN THE PRINCIPAL MATERIAL DIRECTION AT 100 PERCENT DESIGN LIMIT LOAD AT WING STATION Y = 24.00"

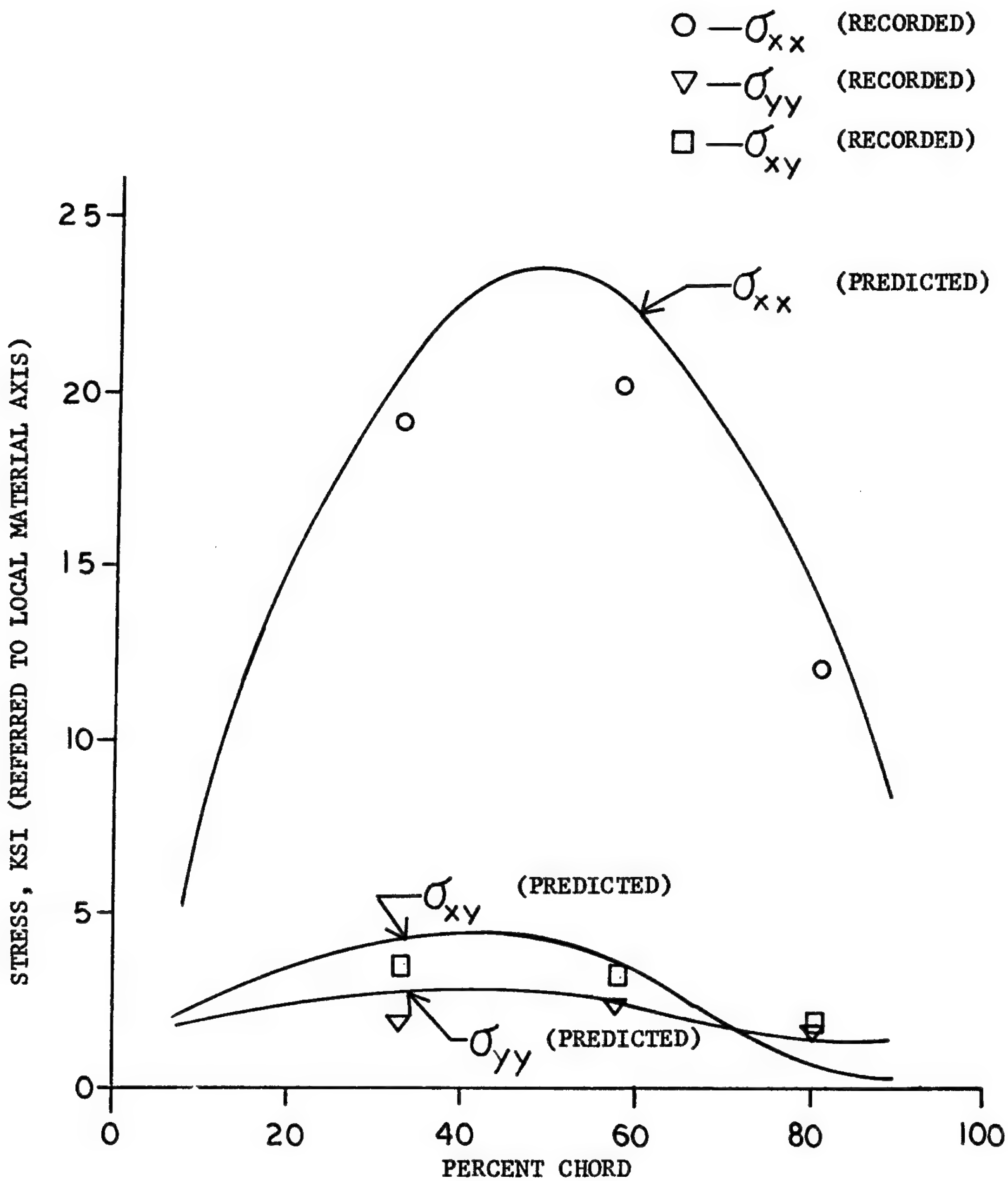


FIGURE 13 - COMPARISON OF PREDICTED AND RECORDED STRESSES IN THE PRINCIPAL MATERIAL DIRECTION AT 100 PERCENT DESIGN LIMIT LOAD AT WING STATION Y = 29.00"

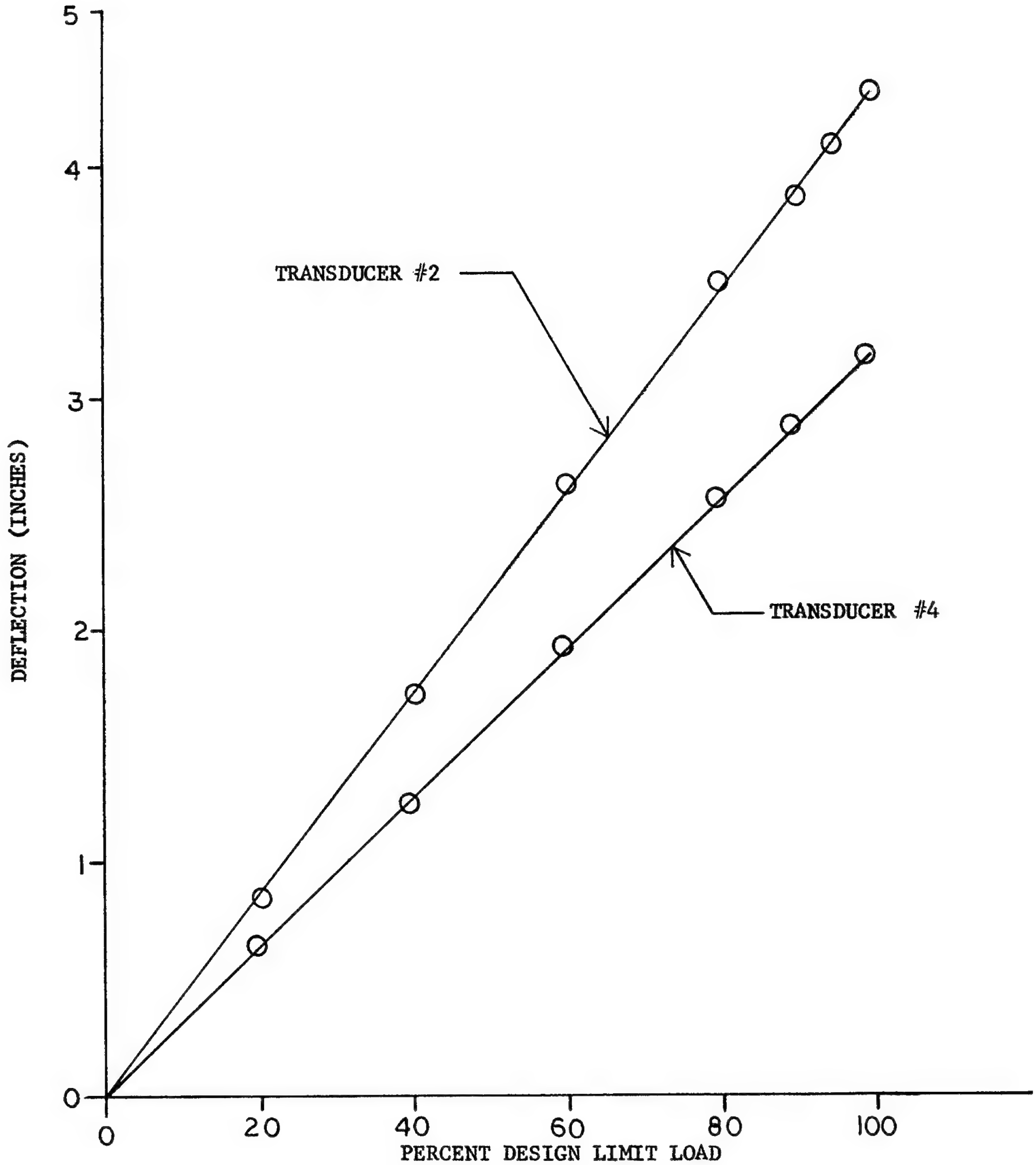


FIGURE 14 - TYPICAL DEFLECTION VS PERCENT LIMIT LOAD TRANSDUCERS 2 AND 4

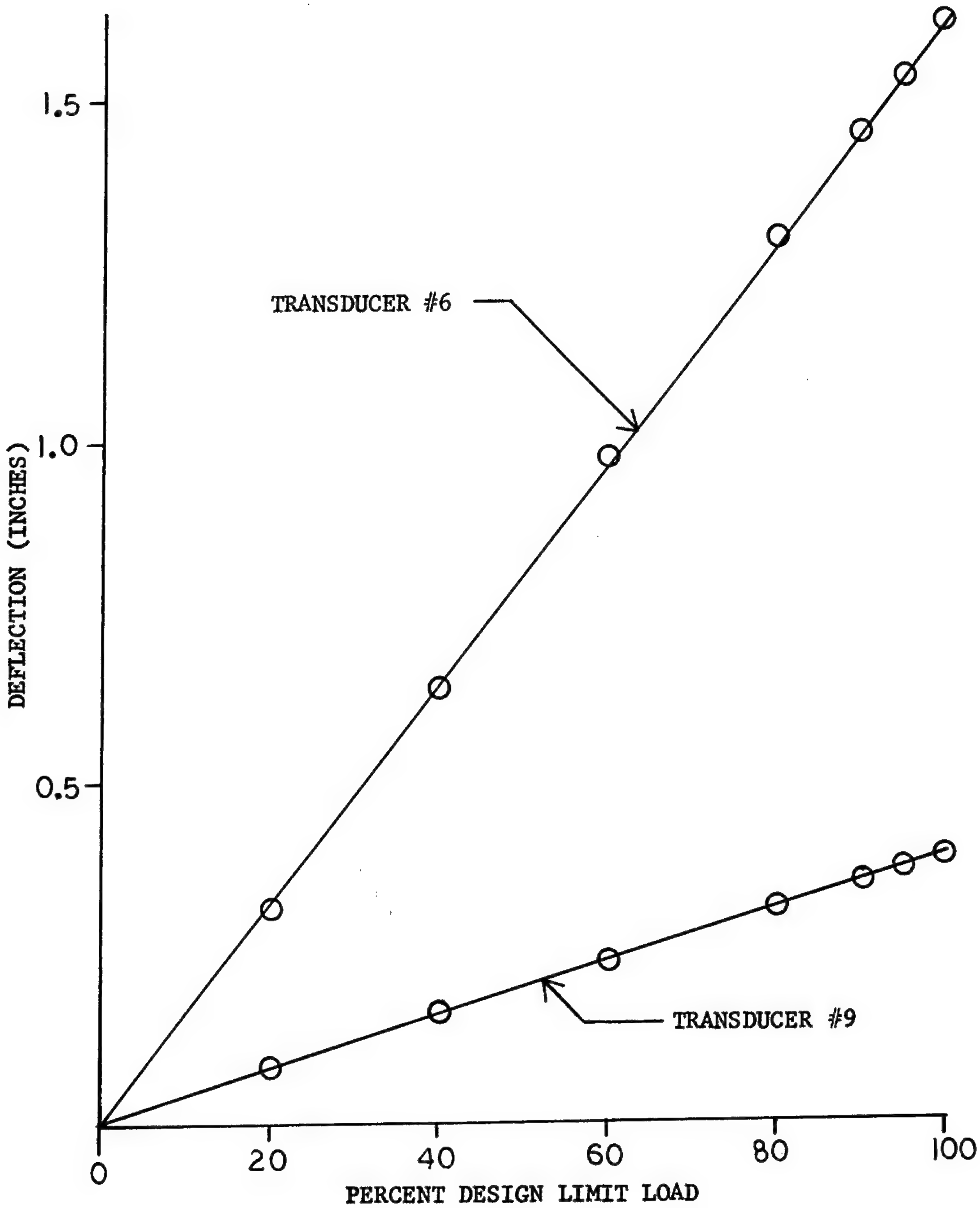


FIGURE 15 - TYPICAL DEFLECTION VS PERCENT LIMIT LOAD, TRANSDUCERS 6 AND 9

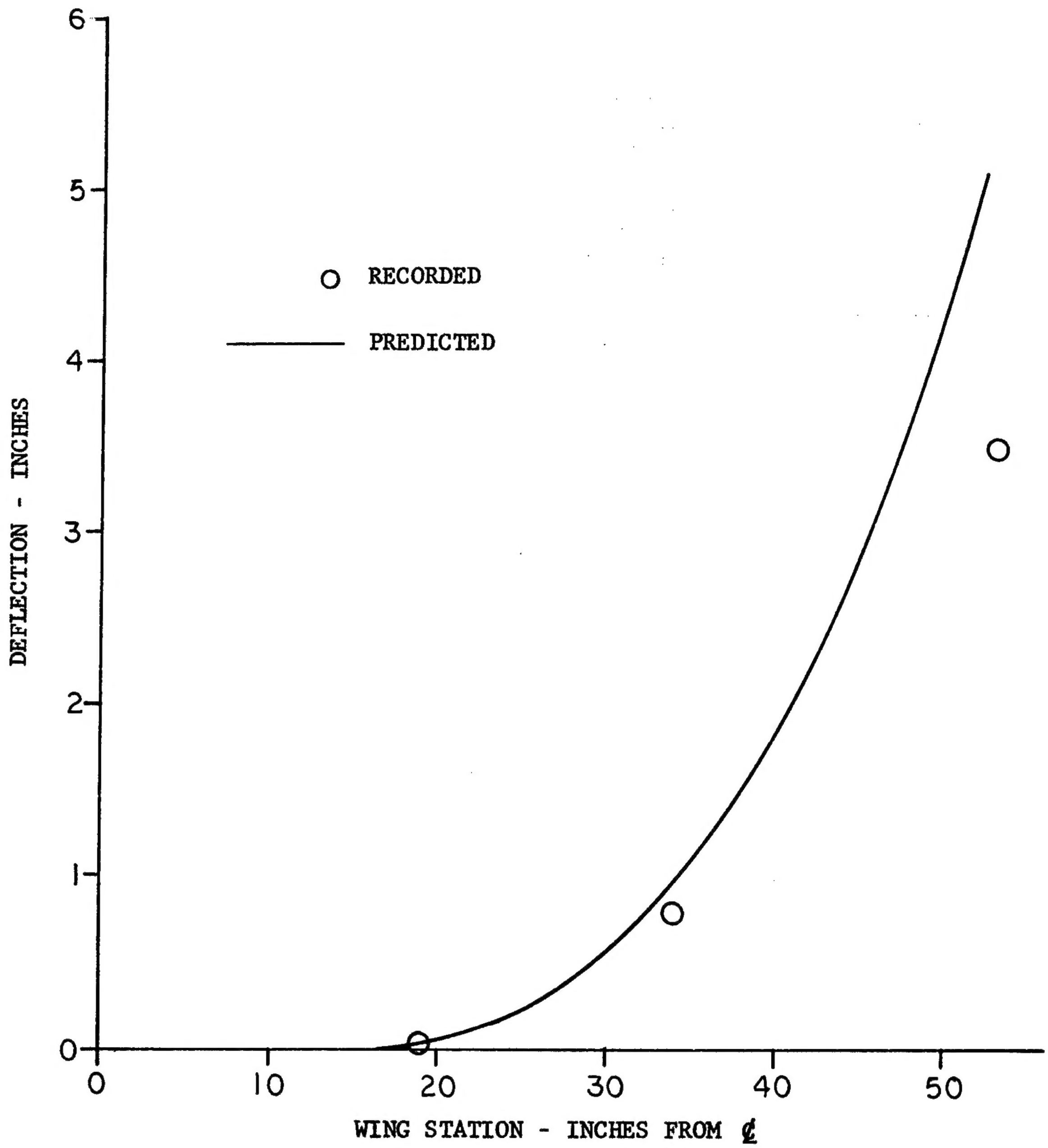


FIGURE 16 - SPANWISE DEFLECTION AT 100 PERCENT DESIGN LIMIT LOAD ALONG WING LEADING EDGE

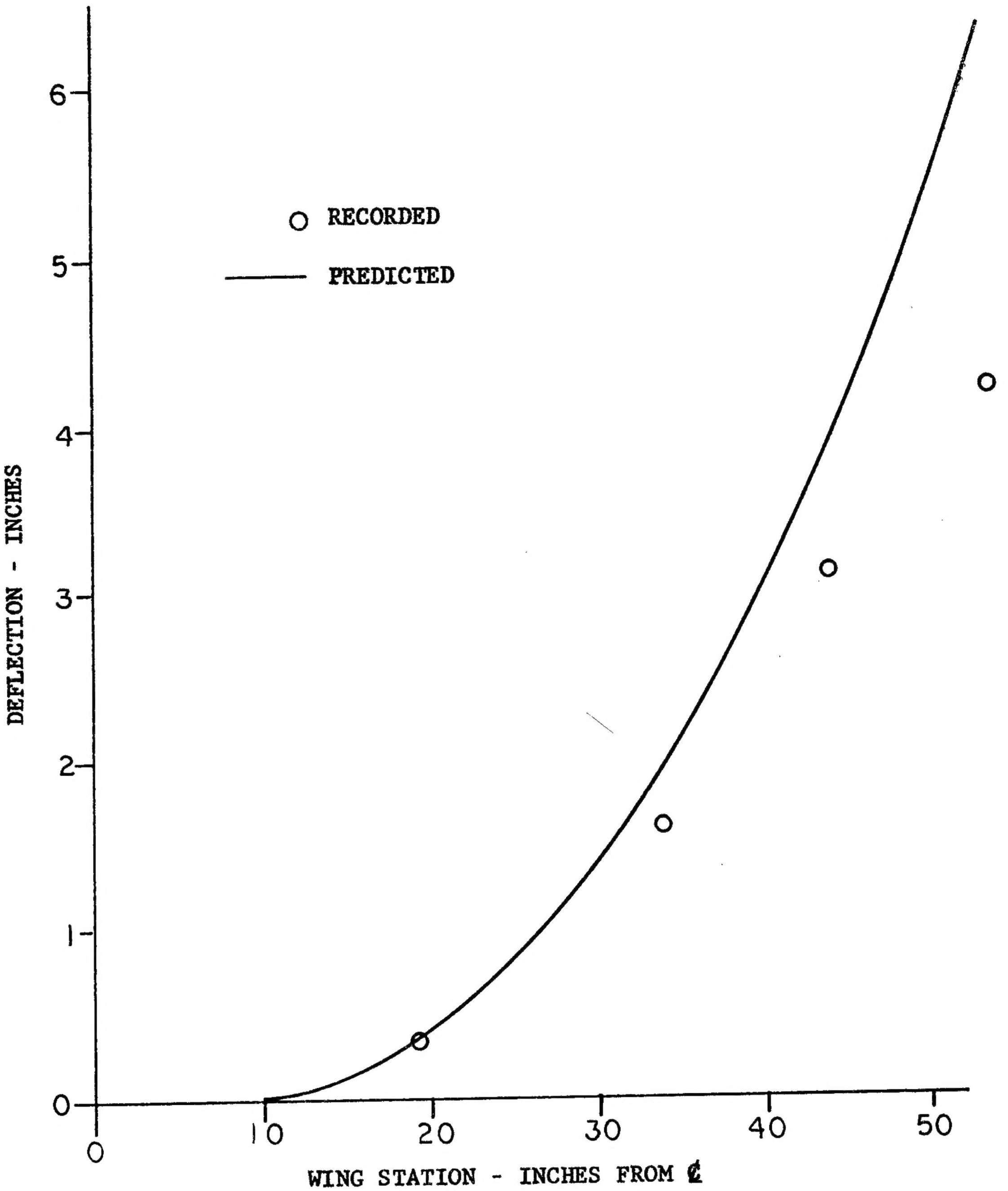


FIGURE 17 - SPANWISE DEFLECTION AT 100 PERCENT DESIGN LIMIT LOAD ALONG 56 PERCENT CHORD LINE

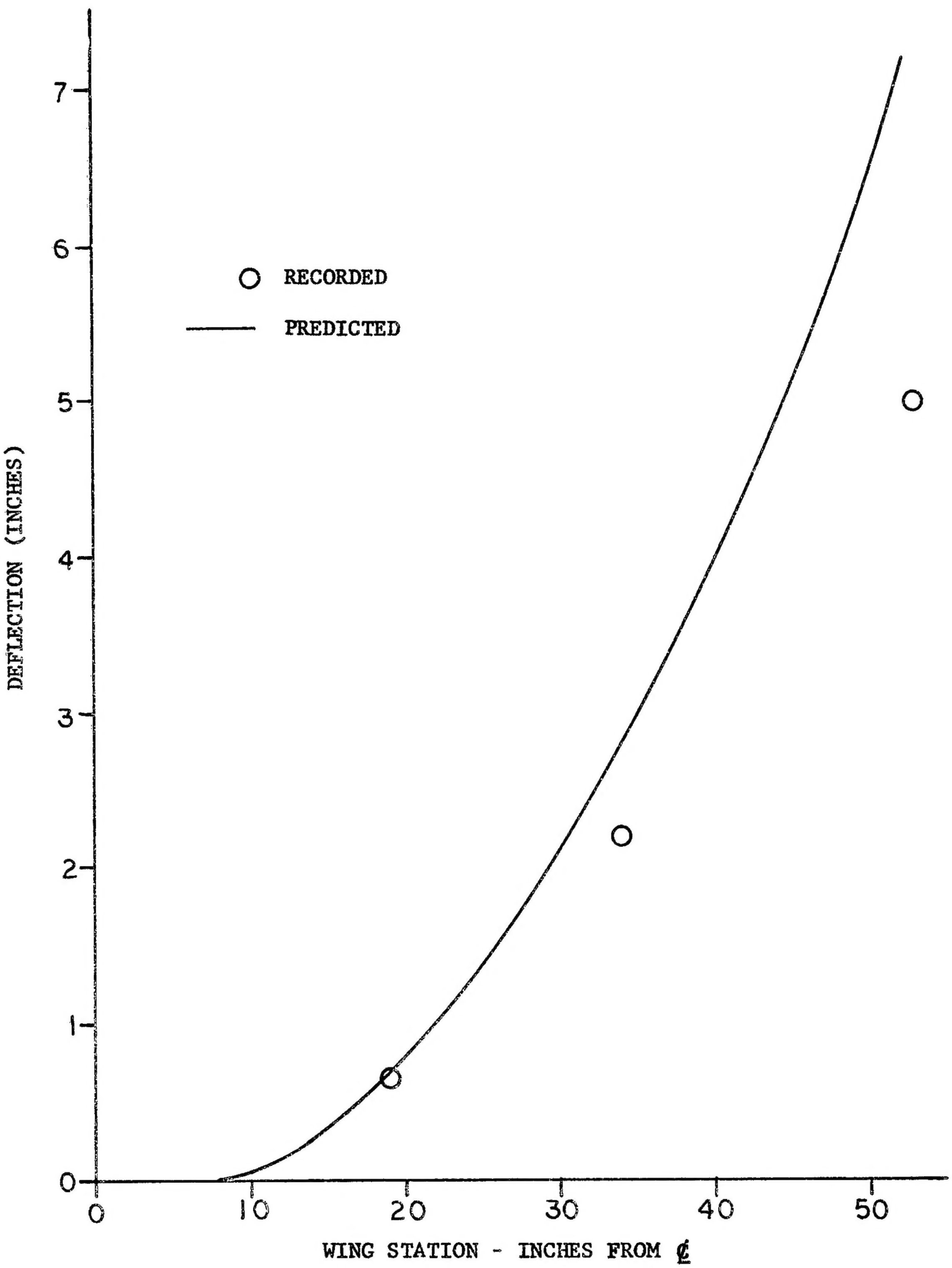


FIGURE 18 - SPANWISE DEFLECTION AT 100 PERCENT DESIGN LIMIT LOAD ALONG WING TRAILING EDGE

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(1 for 303)	
(10 for 3033)	
(1 for 304)	
(1 for 305)	

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